MARS GLOBAL SURVEYOR (MGS) HIGH TEMPERATURE SURVIVAL SOLAR ARRAY

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Abstract

The MGS mission is one of the first major planetary missions conducted under the new NASA Faster, Better, Cheaper guidelines. Ironically, mission requirements make the MGS solar array one of the most challenging designs built for NASA. Not only will the array include silicon and GaAs/Ge panels, but the solar array will be used to aerobrake the spacecraft in the upper regions of the Martian atmosphere. Consequently, even though a mission to Mars is normally typified by cold temperatures, aerobraking imposes a high temperature requirement of nearly 180° C, higher than that experienced by any previous array. The array size is tightly constrained by mass and area. Since the aerobraking occurs early in the mission, it is necessary to subsequently survive up to 20,000 lower temperature thermal cycles. Furthermore, the location of a magnetometer directly on the array structure requires the minimization of circuit induced magnetic moments. This paper provides an overview of the array design and performance. In addition, the high temperature capable design and development will be discussed in detail.

Introduction

The MGS mission is one of the first major planetary missions conducted under the new NASA Faster, Better, Cheaper guidelines. Ironically, mission requirements make the MGS solar array one of the most challenging designs built for NASA. In particular, the solar array will be used to aerobrake the spacecraft in the upper regions of the Martian atmosphere. This will be done to lower and circularize the initially high elliptical insertion orbit, as a method to reduce spacecraft fuel mass. Although the basic technique was proven on the Magellan Venus mapping mission, the conditions for MGS spacecraft vary significantly. The Magellan solar array was designed for a high temperature operating environment and aerobraking was performed only after all original mission objectives were completed. In contrast, MGS will normally be in a low temperature environment, and aerobraking will be performed at the beginning of the primary mission. A failure during this phase will be catastrophic for the mission. After completion of aerobraking the MGS solar arrays will be required to operate for approximately five years and survive over 20,000 additional, less severe, thermal cycles. The MGS array consists of two wings, each with two panels. The outer panels are covered with silicon cells and the inner with GaAs/Ge. Each panel is approximately 1.85 m wide by 1.7 m long.

During the approximately 400 aerobraking orbits, atmospheric drag on the arrays will cause significant heating. In order to avoid damage to the solar cell circuits, the array panels will be oriented so that the rear side will face in the direction of travel during aerobraking. The solar panels will not be generating power during this time. Calculations indicate that the rear surface temperatures may reach as high as mid 180s° C. For this reason, all wiring and connectors will be kept on the panel front surfaces. The front surface temperatures are expected to be approximately 30° C cooler than the rear. Early design considerations indicated that during safe mode, which would occur if there were a spacecraft anomaly, the arrays could face forward during an aerobraking orbit. Consequently, it was necessary to design the entire array for a worst case temperature of approximately 180° C.

In addition to the stringent high temperature requirements, the array mass and area were tightly constrained so that the array design required the use of lightweight components where possible. This included the panel honeycomb substrate which consisted of a low density aluminum honeycomb core with thin composite face sheets. The array area was tied into not only the mass constraint, but also aerobraking drag conditions which balanced heating against drag force. With the need for a fairly high overall array power capability (Table 1) and an additional constraint of cost, the final design resulted in the use of both silicon and GaAs/Ge solar cell covered panels. The former cell type helped keep the costs and mass low, and the latter provided a greater power output. Since the two cell types exhibit different performance during the mission, due to different temperature coefficients and radiation behavior, circuit design had to handle the complex power change throughout the mission. The critical mission design point occurs approximately 2 years into the mission rather than at end of life

Another major array design driver was imposed by the location Of primary science equipment. a magnetometer, on the end of the outer panels, adjacent to the electrical circuits. Combined with the limited panel surface area available for wiring, necessitated a very creative circuit layout and wiring scheme to minimize any magnetic moments Reduction of the array magnetic field is especially critical in view of the weak Martian magnetic field (if any) A unique aspect of the layout was the real time circuit. design conducted jointly by the magnetometer principal investig ator and Spectrolab personnel [1] A final driver was imposed by the array delivery date, which imposed a total cycle from program start to delivery of 14 months, considerably shorter than the Magellan arrays, for example In view of the array complexity end limited time, a team consisting of representatives from Lockheed Martin, Spectrolab, and JPL was assembled to work closely together in order to meet the mission requirements. As will be shown in the following sections, this industry Government team approach load to the successful delivery of the MGS arrays, on schedule and cost, effectively meeting technical and programmatic challenges.

	Outboard	Inboard	Array
Begin of Mission	468 watts	583 watts	2.10 kW
Mid Design Point	151 watts	187 watts	(1.68 kW
End of Mission	128 watts	177 watts	[1.61 kW

Table 1. MGS Power f {requirements at 32 volts L oad

Design for Aerobraking

During aerobraking, the hottest areas will be on the panel rear side. For this reason, all components were attached to the front side. Considerable design effort was exerted to ensure all components could be positioned while maintaining a high solar cell packing factor to meet power requirements and accommodate all wróng. Each material and process proposed for the array was investigated for its ability to survive multiple exposures to 200°C Materials for which specified service temperatures were less than 200°C were subjected to engineering tests to show that they could survive the specified environment. The selected method of interconnection was welding However, the following joint interfaces would be soldered

- Lead wires to circuit termination tabs
- Lead wires to terminals
- Terminals to terminal boards
- Diodes Lo terminals
- Diodes to cell tabs
- Splices (diode wire, wire-wire]
- Replacement tabs to solar cells
- Interconnects to replacement solar cells

A development pr ogram was initiated to select and qualify a solder and soldering process Which would ensure that all the above joint interfaces would survive the specified environment The development program is described in the next section. The solder candidate of choice selected for the array was Sn 95/Ag 5 (M.P. 226°C) Table 2 lists the selected materials and their specified maximum operating temperature. Materials marked with an asterisk (*) were those for which there was a high temperate survivability concern.

Component	Material	Max. Temp.
Solar Cell	Si, GaAs/Ge	450°C
Inter connects	Silver-Plated Moly	960°C
Coverglass	Ceria Doped Microsheet	>1,000°C
Coverglass Adhes	DC 93 500	315°C
Solar Cell Adhesive	CV 2568	450°C
Terminal Board*	Type GFN Copper Clad Fiberglass	130"C
Terminal Board Adhesive*	DC6 1104	200°C
Wire Insulation	MIL-W-22759	300°C
Wire	Copper	1085°C
Connectors	MIL-Cr?430El, type B	200°C
Grounding Adhes *	56C, Cat 9	135°C
Solder*	Sn95/Ag 5	226°C
Diodes*	MIL-S 19500	17!i"c
Insulation 1 ape*	F224	155*C

1 able 2 ARRAY MA1 t RIALS SURVIVABILITY TEMPERATURE

Due to the limited time and funding available, it was clear that only a limited development effort could be carried out Clearly, there was insufficient time to flight qualify ony process so that heritage or slight modification of existing processes was all that could be reasonably expected Following a review of numerous solder candidates, four high temperature scolders wore selected based on melting points and suitability for solar cell assembly Three were tin/silver solder with silver percentages of 3.5,4, and 5 by weight The fourth candidate was a tin/lead material with 2% silver. These materials had minimum melt temperatures of 221° C. Samples of each array scolder joint were made using each of the solders, Standard temperature controlled tips were used and tip temperatures were varied A standard array assembly solder flux was also utilize The ease of flow and solder fillet condition were observed It was noted that the most difficult joint to produce was the terminal to terminal board assembly. It was difficult to flow the solder since the board provided an efficient heat sink The flow was improved by either switching to larger solder tips or preheating the terminal board All solder candidates produced adequate joints which met the criteria of NHB5300.4. This was important as it meant that the existing inspection criteria could be retained The best visual joints were achieved with 95 Sn/5 Ag and this material was generally easiest to flow.

Based on these results, a development coupon was assembled in which all possible solder joints wore included in a flight representative configuration. Other materials were included where manufacturers data indicated that the materials should not survive 200° C (Table 2). The coupon consisted of two strings of seven cells in series. These included welded and repair process solder bonds. A terminal board assembly, with blocking diode was fabricated and redundant bypass diodes were attached to each string Three cells were bonded over Kepton P224 tape patches to assess the impact of exceeding the tape maximum recommended service temperature. Additional P224 patches were placed external to the cell circuits for visual examination of any possible delamination. 95/5 solder was used for all bonds Finally 20 awg ground wires were bonded into holes drilled in the coupon using 56C with catalyst 9. The coupon material was representative of the MGS substrate.

The coupon was then subjected to thermal shock and thermal cycling tests in a GN_2 atmosphere. The thermal shock consisted of 8 cycles, -10° C to 200° C The temperature rate of change was in excess of 30 C per minute with no dwell at the extremes The thermal cycle test consisted of 300 cycles, -145" C to 148° C at a nominal rate of 25° C/ minute with a one minute dwell at each temperature limit. A comprehensive electrical and visual inspection was performed before and after testing. Electrical degradation was within the normal experimental error indicating that the assembly processes survived the high temperatures.

There were no P224 delaminations [either under replaced cells or directly placed on the Kapton insulation] The terminal board had no defects and did not. delaminate, the grounding adhesive showed no sign of degradation and the diodes remained in tact These materials were therefore considered worthy of use in the qualification panel and flight designs. The only material change made in the course rrf the program was to substitute Type GFG copper clad fiberglass (rated to 180°C] for the type GFN noted in Table 2. Some visual anomalies were noted Subsequent examination revealed that one solder joint. had been improperly wet initially rind two others had evidence of handling damage Of greater concern was the appearance of cracks in some of the solder fillets, although no separation was noted A potential cause of the cracks was possible contamination from lead during iron tip tinning Lead contamination can lead to lower melting temperature and reduced joint mechanical strength. It was felt that the use of a dedicated high temperature solder assembly area with new tooling would prevent contamination during flight assembly.

At this time, additional review of the solder properties indicated that although the solder was kept below the melting point., its mechanical strength, even for noncontaminated bonds, was low at temperatures above approximately 130° C. Due to this, a decision was made to reduce any mechanical stresses at the assembly joints In particular, attachments (patent pending] were provided on the end tabs to mechanically hold the soldered wires. The attachments were developed under Spectrolab IR&D in support of the MGS mission Similarly, all diode attachments included hooking the lead wires together prior to soldering and sleeving with shrink tubing A sketch of this is shown in figure 1. In this manner. mechanical loads at the bond would be removed from the solder during the high temperature aerobraking For mission thermal cycles after aerobraking, the maximum solder temperatures would remain below 60° C and high solder strength would be available. The mechanical support would provide fen- an additional safety factor during the worst conditions

Low Temperature, Low Intensity Operation

'The MGS solar array is required to provide power in Martian orbit at illlint.cnskyof(13'7 suns and operating temperature of -5*C (silicon) and + 7 °C (GaAs/Ge). Scalar cell photo current is directly proportional to solar intensity and can easily be calculated. In order to predict the array performance in Martian orbit, it was important to define the voltage loss due to the solar intensity at Mars and the low intensity temperature coefficients for Current and voltage for both types of solar cell As soon es MGS CICs were available, 13 of each cell type were subjected to electrical performance testing at 0.37 suns and temperatures between -80°C and + 75-C. Table 3 shows the low intensity temperature coefficients for each cell type while Table 4 shows the voltage intensity coefficient es a function of temperature 10 predict array performance, the 28°C load voltage intensity coefficients (-.954 for Si, -.960 for GaAs/Ge) were used in conjunction with the low intensity current voltage temperature coefficients It is interesting to note that the low intensity voltage temperature coefficient for silicon is larger than the 1 sun value, while the GaAs/Ge voltage temperature coefficient appears unaffected by reduced intensity.

	Si	GaAs/Ge
V_{mp} (mV/° C)	-2.25	-1.97
V_{oc} (mV/° C)	-2.39	-:'.02
I_{sc} (μa/cm²/° C)	+ 13.$1	+8.1
Imp (μa/cm²/° C)	+ 8.6	+5.9

Table 3 LOW INTENSITY TEMPERATURE COEFFICIENTS (-80° C to +28° C)

	Si	GaAs/Ge
-80°C	.993	.963
-5°C	.975	.966
+7°C	.967	.966
+28°C	.954	.960
+75°C	.945	.956

Table 4 L(MD VOLTAGE INTENSITY COEFFICIENTS vs TEMPERATURE

Panel Design

The panel design was severely constrained by the aerobraking requirement that all components be on the front side, the magnetic requirement of maximum 0.6 nT [dynamic field) and the power requirements at BOM (beginning of mission), MDP (mid design point] and EOM (end of mission). The outboard panel accommodates a magnetometer at the outboard edge and as such has the largest potential impact on the array static magnetic field. Outboard panel power is provided by five circuits 01 BSFR silicon solar cells, subdivided into five strings each of 67 cells series (Figure 2). The cell size is 2.57 cm x 5.88 cms. In general, each string is arranged with its negative termination at the lower edge of the panel and its positive termination at the upper edge of the panel. Each string has a tab attached to the rear of the 37th cell in series to act as a shunt tap. The positive end shunt tap lead wires are routed to the lower panel edge down a cell gap adjacent to the string. This minimizes the net magnetic moment for the string. Strings are laid out track to beck along the panel so that magnetic moments cancel.

In each magnetically canceled string pair, the contribution to magnetic field at the magnetometer of the outboard member of the pair is slightly greater than the inboard member of the pair. To compensate for this, the shunt and positive lead wires, for string 2 of the fourth circuit (being the inboard member of its pair) are routed adjacent to the next inboerd pair [strings 3 and 4 of circuit 4] to provide an extra large current loop. The current loop provides a magnetic field contribution at the magnetometer which balances, Out the surplus field Contribution from the outboard member of each string pair. The exception to the general rule of string layout on the outboard panel is circuit 5 - the one nearest the magnetometer. The panel narrows toward the outboard edge and cannot accommodate a full string length. Also, there is room for only three substrings in this area of the panel. String 1 of the fifth circuit is thus laid out in two substrings; one of 37 cells series the other of 30 cells series. The shunt wire is connected to the negative termination of the shorter string. String 2 of the fifth circuit, is laid out in a similar configuration, except that the shorter string is located at the inboard edge of the panel around the panel hinges. The outgoing and return wires from the inboard edge are twisted together to produce no net field. Because of their proximity to the magnetometer, each of the three substrings are magnetically self cancelled by routing positive wiring down each side of each string. Compensation wire loops are provided for each substring to make one of the current paths more resistive and hence transport slightly less current then its counterpart. In this way, the net magnetic field at the magnetometer can be minimized,

The five strings constituting a circuit are paralleled together at a terminal board located along the lower panel edge. This board also supports the by pass diodes for each circuit. Each haff of the circuit (to the shunt tap point) is protected by three parallel bypass diodes. Three diodes are used to provide adequate de-rating and redundancy in the event of shallowing or cell breakage, The positive and return wiring for each of the five circuit terminal boards is routed along the lower panel edge to another terminal board on the outer panel inner edge. This board supports redundant blocking diodes for each circuit and a common negative bus, It is redundantly wired to the power output connector also located along the inner panel edge. The shunt wires are rootad directly from circuit terminal boards to connector. The outboard panel also has two temperature sensors and two groups of four cells series wired to a telemetry connector on the inboard panel edge.

The inboard panel accommodates a sun sensor on the upper outboard edge. Panel power is provided by six circuits of GaAs/Ge solar cells, subdivided into eight strings of 38 cells series (Figure 3]. The cell size is ? 24 cm x 6.08 cms. In general, each string occupies half the panel width with the negative termination along a panel edge and the positive termination along the panel center line. Each string is broken after the 17th series cell to allow for shunt tap. The shunt taps for adjacent strings in the same circuit are paralleled. Each circuit has foor of its strings in the upper haff of the panel and the other four in the lower half. In this way the panel has almost perfect symmetry and therefore almost perfect magnetic cancellation. The symmetry is not perfect because the outboard edge of the panel can only accommodate three strings due to the spaces occupied by sun sensor and panel hinges. As a result, the opposing strings of each circuit ore staggered by one row, The inboard edge of the panel narrows and panel symmetry is lost in this area since it can only accommodate three strings laid adjacent to each other. Two of these strings are connected to the lower panel half, while the other is connected to the upper panel half.

The negative string terminations are paralleled along the upper and lower panel edges in groups of four and the redundant lead wiring for each group is routed to a terminal board located at the lower edge of the narrow inboard section of the panel. The positive string terminations are paralleled along the panel center-fine in groups of eight and the redundant lead wiring for each group is routed along the central avenue of the panel down to the test, end board The shunt taps for the upper and lower halves of each circuit are spliced together in the central avenue and the six shunt wires are then routed directly to the connector. The terminal board supports redundant blocking diodes for each circuit and a common negative bus. It is redundantly wired to the panel power connector on the lower panel edge. The GaAs/Ge cells are more sensitive to degradation due to reverse bias than the silicon cells. Consequently, by-pass diodes are required approximately every four cells to provide adequate protection against shadowing The by pass diodes are attached via splices and wire leads to tabs emerging from the rear side of every fourth or fifth cell in series and to the string termination tabs and shunt taps.

Qualification Program

In order to gain confidence that the flight array design would survive the! aerobraking environment, a qualification panel was built of flight representative components. The panel consisted of a silicon circuit of five strings laid out identically to circuit 5 of the outboard flight panel, a circuit terminal board with six bypass diodes mounted in flight configuration used to parallel the five silicon cell strings into a circuit; a GaAs/Ge circuit of eight strings laid out in perfect symmetry across the panel center line by pass diodes every fourth or fifth cell to provide shadow protection for each GaAs/Ge string, a terminal board with two pairs of redundant blocking diodes two flight like thermistors; a power connector a telemetry connector and appropriate wiring to connect all the components The qualification panel had an additional 1 072 glass platelets bonded to it to simulate the mass of the missing balance of flight. panel components High temperature solder was used throughout for all joints that were neither welded nor crimped Two cells were intentionally broken in each circuit and replaced using high temperature solder. The panel was subjected to a 3 minute acoustic test to an overall sound level of 143.6 dB Subsequently the panel was placed in a vacuum chamber and thermal shocked between -I O-C and +190˚C for 200 cycles and thermal cycled between -145˚C and + 100˚C for 10 cycles. During one of these cycles, the hot. extreme was increased to 165˚C Finally the panel was held in vacuum at 'I OO˚C until the total elapsed time for thermal cycle and bake out was 168 hours. Despite the ri, 101 ous nature of this test program, only the normal attrition due to infant mortality was observed 1 here was no delamination and no joint failures.

Flight Panel Performance and Test

The success of the qualification panel in surviving the aerobraking environment permitted the initiation of flight panel assembly The GaAs/Ge solar cells were fabricated using state-of-the art MOVPE technology After front side weld interconnect attachment, the cells were filtered with 5 mil ceria doped microsheet coverglass and tested at constant current. The electrical distribution of the CIC population resulted in an average efficiency over the build of 19.2% The silicon cell build was also successful, producing an average efficiency over the CIC population of 14.8% The CICs of each cell type were series welded into strings and bonded to the graphite panel substrates by vacuum bag technique After assembly, the flight panels were arranged in a flight wing configuration, with a magnetometer attached to the outboard edge of the outboard panel 1 he two panels were energized using a power supply to a current level equivalent to the flight condition at Mars The magnetic field of each panel and the two panels combined was measured by the magnetometer. The results are reported in another paper at this conference. The measured magnetic fields were acceptably close to the specified maximum field requirement and as such represent the lowest magnetic fields ever produced by a solar cell array

1 he acceptance testing of the flight panels included a 1 minute acoustic test to an overall sound level of 139,6 dB, six thermal vacuum cycles between -1 25˚C and +80˚C with the hot extreme during one of these cycles increased to +165*C. Finally the panels were held at 80˚C until total elapsed time for thermal cycle and bake out was 68 hours The as-shipped electrical performance of all four panels exceeded specification by an average 3.3% The details are shown in Table 4.

		Requirement	Actual	Delta
GaAs/Ge	001	18.22	18.83	+3.380/0
	002	18.22	1 8.8?	+3.28%
Si	001	14.63	15.04	+2.84%
	[K.)"	14.63	15.16	+3.66%

Table -1 ELECTRICAL PERFORMANCE (CURRENT @ 32V)

Allowing for losses in wiring and blocking diodes, the on panel efficiencies at the maximum power point were 18.9% (GaAs/Ge) and 14.6% (Si) Finally, the inboard panels were shipped al more than 12% under and the outboard panels 4% under the specified maximum add on mass 1 he details are shown in Table 5.

		Requirement	Actual	Delta
GaAs/Ge	001	13.45 lbs	11.74 lbs	-12.7%
	002	13.45 lbs	11.84 fbs	-12.0%
Si	001	8.60 lbs	8.2 lbs	- 3.9%
	002	8.60 lbs	8.?2 lbs	4 . 4 %

"Table 5 MGS ADD ON MASS

CONCLUSIONS

The MGS solar arrays are completed flight acceptance tests and were delivered to Lockheed Mar-tin for spacecraft integration during the past Summer. The launch is scheduled for November, 1996. MGS is the first in a planned series of spacecraft that form the Mission to Mars program. It is one of the first missions to be conducted under the NASA Faster, Better, Cheaper guideline: The MGS solar array technical requirements ant) schedule were extremely demanding, Probably more than eny previous NASA mission. These included vary high temperature survival tile}, extensive thermal cycling, high specific power (power/mass) very low total magnetic moment, use of silicon and GaAs/Ge cell circuits on tho same array (different panels], and a performance schedule between a half to a third that of previous NASA PV systems. By working together as a team, Lockheed-Martin, Spectrolab, and JPL were able to develop real time solutions to any problems. As a result, the MGS solar array met or exceeded all requirements and successfully completed qualification and acceptance testing and met the necessary delivery date. The fourteen month program duration is a faster array build cycle then has been typical for single nonrecurring arrays. The performance of the array be ttered the primary requirements for mass, power, and magnetic momont And the overall cost was kept to the original projected value by minimizing design changes during the program.

A word of caution needs to be expressed however. NASA plans call for a significant increase in the number of Planetary and Interplanetary missions. The majority of these will be solar powered. These missions will extend the range of photovoltaics, environments end operating conditions hitherto not experienced. Those will include higher and lower operating temperatures, and particulate radiation under low operating temperatures, to list a few. Most array designs have evolved from Earth orbiting systems and es seen with the MGS array, significant new requirements can be expected from the future missions. The repid spacecraft build cycles and limited funding available may not always result in a successful mini-development program as discussed here, It will not always be possible for the array manufacturer to apply IR&D to an ongoing production program. NASA missions, even under optimistic scenarios, are likely to remain at less than 5% Of an array manufacturers annual *on. Consequently, it is not clear that sufficient resources end solutions can always be available to the specialized needs of a small program. Ths is an area where it would be worthwhile for NASA to consider developing specific technologies for future missions.

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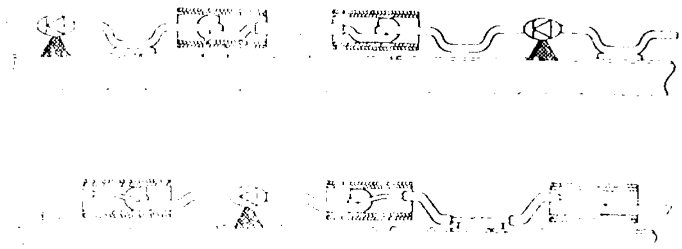

Figure 1. Bypass Diode attachment

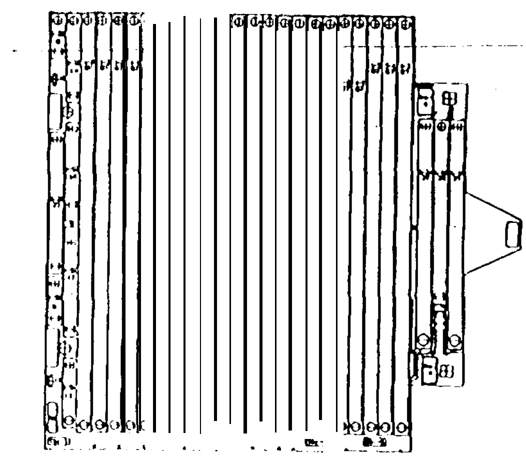

Figure 2. Outboard panel (MAGNETOMETER AT RIGHT)

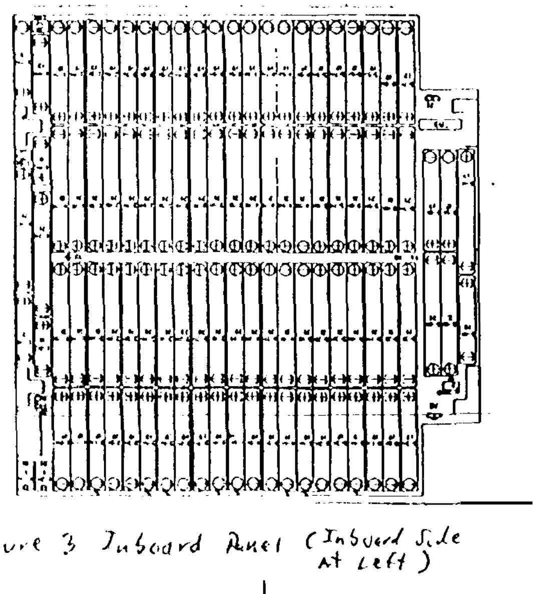

Figure 3 Inboard Panel (Inboard Side at Left)